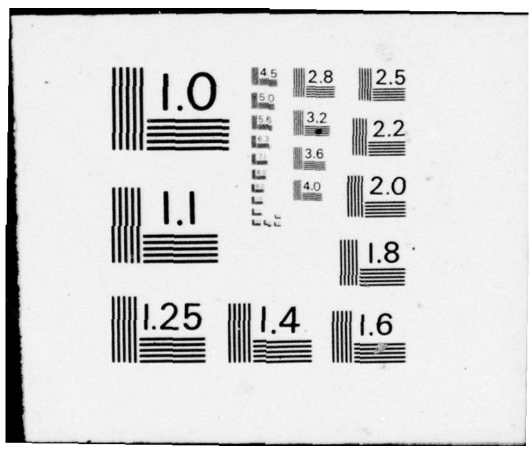
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MORE EFFECTIVE AIRCRAFT STABILITY AND CONTROL FLIGHT TESTING THROUGH USE OF SYSTEM IDENTIFICATION TECHNOLOGY

by

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PREFACE

This Technical Memorandum presents an overview of the system identification program that was initiated at the Naval Air Test center in 1970. This program has resulted in an advanced system identification capability that has a wide spectrum of applications to aircraft flight testing. An overview of maximum likelihood parameter identification and applications to aircraft stability and control flight testing are presented.

This Technical Memorandum was prepared for presentation to the AIAA Systems and Technology Meeting in Dallas, Texas, on 27-29 September 1976. The research performed to develop the technology reported on was conducted under a series of AIRTASK Assignments and Procurement Requests sponsored by the Naval Air Systems Command and Office of Naval Research, respectively. These programs were managed by Mr. Ralph A'Harrah (AIR-53011) at the Naval Air Systems Command and Mr. Dave Siegel (ONR Code 211) at the Office of Naval Research. A research team at Systems Control, Incorporated, headed by Dr. W. E. Hall, developed the computer program SCIDNT.

APPROVED FOR RELEASE

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Abstract

The development of system identification technology was undertaken to provide for more effective aircraft flight testing by reducing the time required to conduct specific tests and/or to provide for a more comprehensive data analysis. F-14A and TA-4J flight test results presented demonstrate that the flight time required to obtain stability and control data can be significantly reduced without loss in accuracy of conventional flight test derived parameters. Presentation of S-3A and EA-6B system identification results demonstrate that this technology can be successfully used to update the aerodynamic data bases of modern jet aircraft from flight test data. These system identification results are compared with wind tunnel data and flight test derived parameters to demonstrate the accuracy of this new technology. Applications of this technology to integrate several areas of aircraft flight testing are discussed.

List of Symbols

Symbol	Definition	Units
a_{z_m}	Measured vertical acceleration	ft/sec^2
a_{z_T}	True vertical acceleration	ft/sec^2
a_{y_m}	Measured lateral acceleration	ft/sec^2
a_{y_T}	True lateral acceleration	ft/sec^2
b	Measurement bias vector	-
D	Matrix relating measurements to control vector	-
F	Matrix of stability derivatives	-
G	Matrix of control derivatives	-
g	Acceleration due to gravity	ft/sec^2
H	Matrix relating measurements to state vector	-
I	Identity matrix	-
I_{XX}	Moment of inertia about roll axis	slug-ft^2
I_{XZ}	Moment of inertia about yaw axis	slug-ft^2
K	Gain	-
K_β	Sideslip vane scale factor	-
K_α	Angle of attack vane scale factor	-
ℓ_Z	Z distance of sideslip vane to center of gravity	ft
ℓ_X	X distance of sideslip vane to center of gravity	ft
L	Rolling moment about X axis	
N	Yawing moment about Z axis	
n_z	Normal acceleration ($n_z = -a_z$)	ft/sec^2
p	Roll rate	rad/sec
q	Pitch rate	rad/sec
r	Yaw rate	rad/sec
s	Laplace operator	-
t	Time	sec
u	Airspeed	ft/sec
x	State vector	-
X	X component of force	lb
y	Measurement vector	-
$\hat{y}$	Estimate of measurement vector	-
Y	Y component of force	lb
Z	Z component of force	lb
$(\dot{\ })$	Time rate of change	sec^{-1}
$(\hat{\ })$	Estimate	
α	Angle of attack	rad
β	Sideslip angle	rad
Δ	Characteristic equation	-
δ_a	Aileron deflection	rad
δ_e	Elevator deflection	
δ_R	Rudder deflection	rad
δ_{sp}	Spoiler deflection	rad
ζ	Damping ratio	
θ	Pitch attitude	rad
σ	Parameter estimate variance or confidence bound	-
τ	Time constant	sec
υ	Measurement vector random error	-
ϕ	Roll angle	rad
ω	Natural frequency	

L_x	Primed partial derivative of rolling moment with respect to x (where x is p, r, β, δa, δR, δsp)	sec^{-1}
M_x	Partial derivative of pitching moment with respect to x (where x is q, α, u, δ_e)	sec^{-1}
N_x	Primed partial derivative of yawing moment with respect to x (where x is p, r, β, δ_a, δ_R, δsp)	sec^{-1}
X_x	Partial derivative of X force with respect to x (where x is q, α, u, δ_e)	sec^{-1}
Y_x	Partial derivative of sideforce with respect to x (where x is p, r, β, δ_a, δ_R, δsp)	sec^{-1}
Z_x	Partial derivative of Z force with respect to x (where x is q, α, u, δ_e)	
C_{K_x}	Non-dimensional partial derivative of K force or moment (L, M, N, X, Y, Z) with respect to state vector x or control vector δ)	

Subscripts

m	Measured
T	True
$(\ )_o$	Trim condition

Superscripts

T	Transpose

Background

The Naval Air Test Center (NATC) initiated a program in 1971 to develop advanced system identification techniques for use in flight testing aircraft. This program has been a coordinated effort among the Naval Air Test Center, Naval Air Systems Command, Office of Naval Research, and Systems Control, Incorporated. The major objective of the development of this technology has been to provide for more effective flight testing by:

a. Improving safety of flight.

b. Reducing cost and/or time associated with design, flight tests, and certification of aircraft.

c. Improving data analysis accuracy.

d. Providing a basis for more comprehensive analysis of mission suitability.

e. Providing for the acquisition of flight test data for use in flight trainers.

f. Providing an accurate data base for the design and/or modification of advanced flight control systems.

To date, the application of system identification to Navy flight testing has been primarily in the areas of stability and control testing and the updating of aircraft aerodynamic data bases for use in system redesign or modification. For example, the determination of the compliance of an aircraft stability and control characteristic with the requirements of Military Specification MIL-F-8785B is a costly and time-consuming facet of aircraft flight testing.(1) However, a considerable portion of the stability and control flight program can be eliminated through use of system identification by using this technology to extract the aerodynamic stability and control derivatives from a limited number of flight tests. These stability and control derivatives are then used to verify the aircraft's compliance with the Military Specification requirements.

Overview of NATC System Identification Program

In order to accomplish the overall objectives of the development of this technology (items a. through c. of the preceding section), a comprehensive system identification capability will be required. In general, a total system identification capability is considered to consist of:

a. Design of experiments (input design).

b. Model structure determination.

c. Parameter identification.

The algorithms that NATC is developing to formulate this system identification capability are:

a. Linear maximum likelihood parameter identification.

b. Nonlinear maximum likelihood parameter identification.

c. Data consistency (state estimation).

d. Instrumental variable parameter identification (real-time parameter identification).

e. Transfer function analysis.

f. Time series analysis.

g. Vehicle dynamics simulation.

h. Model structure determination.

The one key element in the formulation of all of these algorithms is that they have been programmed in a general format such that they can be easily modified for the analysis of any type of system. Item a. is an operational program, and items b. through f. have been installed on the NATC computer system and are currently under evaluation. Items g. and h. are currently under development and are planned for completion by the end of this year.

Parameter Identification Procedure Used in this Analysis

The results presented in this paper are based on a linear maximum likelihood parameter identification computer algorithm (SCIDNT III) currently being used in the analysis of flight test data at the Naval Air Test Center.(2) This program provides for the estimation of:

a. Coefficients of linear systems.

b. Instrumentation system scale factor errors, biases, and lags.

c. Gust time history characteristics.

Thus, if a system is modeled in conventional state-space notation:

State Equation

$$\dot{x} = Fx + G\delta + \Gamma w \tag{1}$$

where:

x is a nx1 state vector

δ is a ℓ x1 control vector

w is a qx1 random process noise vector

F is a nxn matrix of stability derivatives

G is a nx ℓ matrix of control derivatives and corrections for initial out-of-trim conditions

Γ is a nxq process noise distribution matrix

Measurement Equation

$$y = Hx + D\delta + b + v \tag{2}$$

where:

y is a rx1 measurement vector

B is a rx1 measurement bias vector

v is a rx1 measurement random noise error vector

H is a rxn matrix relating the measurements to the state vector

D is a rx ℓ matrix relating the measurements to the control vector

and

n is the number of states

ℓ is the number of controls

r is the number of measurements

and w and v are Gaussian random noises which have zero mean, are uncorrelated, and have power spectral densities Q and R, respectively.

Then, this parameter identification algorithm provides for estimates of the elements which make up F, G, H, D, and B. In addition, estimates are made of the power spectral density of the process noise w and the measurement noise v. This identification algorithm is programmed in a general format such that the state equation and measurement equation can be easily modified. Thus, this program is easily changed to account for any differences in aerodynamics or control system characteristics between airplanes or to model any type of system such as an aircraft's power plant.

In general, system identification is the process of estimating from a given set of input/output data either the required structure of a system model or the parameters of a prespecified model. The general process of parameter identification is illustrated in figures 1 and 2. Figure 1 is an overview of the elements in parameter identification, and figure 2 is a schematic of the iterative nature of the algorithm. As shown, the parameter estimation procedure begins with the system or aircraft being disturbed from some initial condition by a pilot input. The aircraft response to this input is then recorded on the instrumentation system (in general, this measurement can be corrupted by external disturbances and/or instrumentation noise). This measured flight response is compared with a computed response based on a mathematical model of the aircraft to form a response error. This response error is formulated into a criterion function which is used in an estimation algorithm to update the initial guess of the parameter values. This procedure is repeated until the response error ideally goes to zero, at which point, the 'best' estimate of the aircraft parameters is obtained.

In the maximum likelihood algorithm used, the criterion function is the likelihood function

$$\mathcal{L}\left(\theta/y\right) = P\left(y/\theta\right) \tag{3}$$

which is defined as the conditional probability of the measurements (y) having occurred given the parameter set θ (θ is the set of parameters defined from the system state and included in the feasible set Θ). Thus, for any given set of values of the parameter θ , we can assign a probability of $P\left(y/\theta\right)$ to each outcome y. If the outcome of an actual flight test is y, it is of interest to know which set of values might have led to these observations. In order to accomplish this, the maximum likelihood method finds a set of parameters to maximize the likelihood function such that

$$\hat{\theta} = \max_{\theta \in \Theta} P\left(y/\theta\right) \tag{4}$$

In other words, the probability of the outcome y is higher with parameters $\hat{\theta}$ in the model than with any other values of parameters from the feasible set. The likelihood function used is given by:

$$P\left(y(t_i)/y(t_{i-1}),\theta\right) = \frac{\exp\left\{-\frac{1}{2}\nu^T(i)\,B^{-1}(i)\nu(i)\right\}}{(2\pi)^{M/2}\left|B(i)\right|^{1/2}} \tag{5}$$

or in more convenient log form:

$$\log\left(P(y(t_i)/y(t_{i-1}),\theta\right) = -\frac{1}{2}\left[\nu^T(i)\,B^{-1}(i)\nu(i) + \log\left|B(i)\right|\right] + \text{CONSTANT} \tag{6}$$

where

$$\nu(i) = y(i) - \hat{y}(i) \tag{7}$$

and m is the number of measurements and B is approximated by

$$\hat{B} \approx \frac{1}{N}\sum_{i=1}^{N}\nu(i)\nu^T(i) \tag{8}$$

The optimization procedure used to find the parameter set is the modified Newton Raphson technique

$$\hat{\theta}_{i+1} = \hat{\theta}_i + \Delta\theta \tag{9}$$

where

$$\Delta\theta = -\left\{\frac{\partial^2 \log P(y/\theta)}{\partial\theta_j\partial\theta_k}\right\}^{-1}\frac{\partial \log P(y/\theta)}{\partial\theta_j} \tag{10}$$

and i is the ith iteration and j and k denote the jth and kth parameters.[3]

Applications of System Identification to Determining Stability and Control Flight Test Requirements

The military flying qualities specification, MIL-F-8785B, formulates many stability and control requirements in terms of transfer function characteristics and mode ratios. For example, many dynamic longitudinal and lateral directional characteristics are specified in terms of transfer function numerator (zero) and denominator (pole) root locations (frequency, damping, and time constants). Static stability characteristics such as longitudinal control (elevator position) variation with airspeed are related to the appropriate transfer function evaluated at steady state conditions (the Laplace operator $s = 0$). Specification requirements for the ratios of normal acceleration to angle of attack (n_z/α) and roll angle to sideslip angle (ϕ/β) can be determined using mode ratios evaluated at the correct frequency. Compliance with these types of specification requirements can easily be determined through use of system identification.

If a simplified form of equations (1) and (2) is considered

$$\dot{x} = Fx + G\delta \tag{11}$$

$$y = Hx + D\delta \tag{12}$$

The Laplace transform solution of equation (11) for zero initial conditions is given by:

$$x(s) = (sI - F)^{-1} G\delta(s) \tag{13}$$

where I is the identity matrix. This expression is now substituted into the Laplace transform of equation (12) to obtain

$$y(s) = H(sI - F)^{-1} G\delta(s) + D\delta(s) \tag{14}$$

A general expression for the output to input transfer functions of the system described by equations (11) and (12) is then given by:

$$\frac{y(s)}{\delta(s)} = H(sI - F)^{-1} G + D \tag{15}$$

If the elements in the matrices F, G, H, and D can be estimated from flight test data using system identification techniques, it is easily seen that equation (15) can be used to determine the stability and control specification requirements previously mentioned. (It should be noted that this equation is solved on a digital computer to obtain numerator and denominator roots and classical approximations for these terms are not made.) Specific examples for determining longitudinal and lateral-directional dynamic and static stability characteristics are now presented.

Longitudinal Specification Requirements

The angle of attack to elevator input transfer function is given by:

$$\frac{\alpha(s)}{\delta_e(s)} = \frac{N_{\delta_e}^{\alpha}}{\Delta} \tag{16}$$

where the numerator polynominal is of the form

$$N_{\delta_e}^{\alpha} = K_\alpha(s + 1/\tau_\alpha)(s^2 + 2\zeta_\alpha \omega_{n_\alpha} s + \omega_{n_\alpha}^2) \tag{17}$$

and the denominator polynominal (characteristic equation) is

$$\Delta = \underbrace{(s^2 + 2\zeta_p \omega_{n_p} s + \omega_{n_p}^2)}_{\text{Phugoid Mode}}\underbrace{(s^2 + 2\zeta_{sp}\omega_{n_{sp}} s + \omega_{n_{sp}}^2)}_{\text{Short Period Mode}} \tag{18}$$

Similarly, the normal acceleration to elevator input transfer function is given by:

$$\frac{n_z(s)}{\delta_e(s)} = \frac{N_{\delta_e}^{n_z}}{\Delta} \tag{19}$$

where the numerator polynominal is of the form

$$N_{\delta_e}^{n_z} = K_{a_z} s(s + 1/\tau_{a_{z_1}})(s + 1/\tau_{a_{z_2}})(s + 1/\tau_{a_{z_3}}) \tag{20}$$

Thus, the dynamic longitudinal stability specification requirements that can be determined by evaluating equations (16) through (20) are the phugoid frequency (ω_{n_p}) and damping (ζ_p) and short period frequency ($\omega_{n_{sp}}$) and damping (ζ_{sp}). This is further illustrated by considering a classical approximation to these terms, from which it is seen that once the stability and control derivatives are known, it is a simple matter to calculate these specification requirements.

$$\zeta_p \approx \frac{1}{2\omega_{n_p}}\left\{\frac{M_u(X_\alpha - g)}{M_\alpha} - X_u\right\} \tag{21}$$

$$\omega_{n_p}^2 \approx g\left(\frac{M_u Z_\alpha}{M_\alpha} - Z_u\right) \tag{22}$$

and

$$\zeta_{sp} \approx -\frac{1}{2\omega_{n_{sp}}}(Z_\alpha + M_q) \tag{23}$$

$$\omega_{n_{sp}}^2 \approx Z_\alpha M_q - M_\alpha(Z_q + 1) \tag{24}$$

The normal acceleration sensitivity specification requirement can be obtained by forming the mode ratio (n_z/α) from equations (16) and (19) and evaluating it at the short period root location:

$$\left.\frac{n_z(s)}{\alpha(s)}\right|_{s = -\zeta_{sp}\omega_{n_{sp}} \pm j\omega_{n_{sp}}(1 - \zeta_{sp}^2)^{1/2}} = \frac{N_{\delta_e}^{n_z}}{N_{\delta_e}^{\alpha}} \tag{25}$$

Static longitudinal stability requirements in terms of elevator position gradients with airspeed can be determined by evaluating the airspeed (u) to elevator input transfer function at steady state conditions.

$$\frac{u(s)}{\delta_e(s)} = \frac{N_{\delta_e}^{u}}{\Delta} = \frac{K_u(S + 1/\tau_{u_1})(S + 1/\tau_{u_2})}{\Delta} \tag{26}$$

Evaluating (26) at $s = 0$

$$\left.\frac{u(s)}{\delta_e(s)}\right|_{s=0} = \frac{K_u(1/\tau_{u_1})(1/\tau_{u_2})}{\omega_{n_{sp}}^2 \omega_{n_p}^2} \tag{27}$$

which is equivalent to

$$\left.\frac{u(s)}{\delta_e(s)}\right|_{s=0} \approx \frac{-(-Z_\alpha M_{\delta_e} + M_\alpha Z_{\delta_e})}{(M_\alpha Z_u - M_u Z_\alpha)} \tag{28}$$

Thus, equation (28) can be interpreted as the gradient of airspeed with elevator position during a static longitudinal test.

Lateral-Directional Specification Requirements

In the lateral-directional axes, MIL-F-8785B sets new requirements for Dutch roll damping (ω_{n_d}) and frequency (ζ_d), spiral mode ($1/\tau_s$), roll mode ($1/\tau_R$), and roll rate. In addition to the updated military specification requirements in the lateral-directional axes, there are also new parametric requirements in the detail specification for the S-3A and F-14A airplanes. These new requirements are in the form of the Dutch roll coupling parameter $\left(\frac{\omega_{n_\phi}}{\omega_{n_d}}\right)$ and the Dutch roll excitation parameter $\left(\frac{K_d}{K_{ss}}\right)$. These new specification requirements in the lateral-directional axes are difficult to determine accurately because the effects of the spiral, roll, and Dutch roll modes cannot be easily separated using conventional data techniques. However, these new requirements can easily be determined if the roll rate to aileron input transfer function is evaluated using estimated stability and control derivatives.

$$\frac{p(s)}{\delta_a(s)} = \frac{K_\phi s(s^2+2\zeta_\phi\omega_{n_\phi}s+\omega_{n_\phi}^2)}{(s+1/\tau_R)(s+1/\tau_s)(s^2+2\zeta_d\omega_{n_d}s+\omega_{n_d}^2)} \tag{29}$$

Compliance with the specification requirements can be determined from the estimated transfer function parameters ω_{n_d}, ζ_d, $1/\tau_s$, $1/\tau_R$, $\frac{\omega_{n_\phi}}{\omega_{n_d}}$ and $\frac{K_d}{K_{ss}}$. This is illustrated by considering a classical approximation to the roll mode time constant

$$1/\tau_R \approx -L_p' + \frac{L_\beta'\left(N_p' - \frac{g}{u_0}\right)}{N_\beta'} \tag{30}$$

In order to determine $\frac{K_d}{K_{ss}}$ from these data, the matched transfer function poles and zeros are plotted on a s-plane as shown in figure 3. The term K_d is then determined as the residue measured from the Dutch roll pole and is given by:

$$K_d \Big|_{\substack{\text{Dutch}\\ \text{Roll}\\ \text{Residue}}} = \frac{a\,b}{e\,\omega_{n_d}\omega_d} \tag{31}$$

where a, b, and e are defined in figure 3. The term K_{ss} is the steady state residue and is measured from the origin, assuming that $1/\tau_s = 0$.

$$K_{ss} \Big|_{\substack{\text{Steady}\\ \text{State}\\ \text{Residue}}} = \frac{\omega_{n_\phi}^2 \tau_R}{\omega_{n_d}^2} \tag{32}$$

Thus, K_d/K_{ss} is determined as the ratio of equations (31) and (32).

Flight Test Results

Reductions in Flight Testing Using System Identification

System identification technology can be used to reduce the flight time required to obtain data for determining compliance with stability and control specification requirements. This is accomplished by using this technology to extract from a limited number of flight tests the stability and control derivatives which are then used to determine compliance with specification requirements which normally require multiple tests at each trim flight condition. In order to accomplish this objective, it was necessary to conduct a flight program to determine the optimal flight inputs for parameter identification analysis.[4] The result of this research effort and additional follow-on work has been the selection of a sequential aileron-rudder doublet for lateral-directional stability analysis and an elevator doublet or sine wave for longitudinal short period analysis (a sine wave appears to have no advantages over a doublet input for linear analysis). Parameter identification analysis of the phugoid mode requires use of the conventional test technique.

The use of these 'optimal' inputs to reduce flight tests is demonstrated in table I. As shown, system identification techniques can be used to reduce the total number of maneuvers required in determining compliance with the longitudinal and lateral-directional stability and control specification requirements previously noted by a factor of 3 at each individual trim point (a reduction from 9 maneuvers to 3 maneuvers). This reduction in the number of maneuvers required results in a savings of approximately 75 percent in flight test time (based on TA-4J flight tests conducted by the U.S. Naval Test Pilot School). The TA-4J was then used in a test designed to provide flight data for determining the aircraft's characteristics over an airspeed range at one altitude. This consisted of accelerating the aircraft from 165 knots indicated airspeed to the maximum level flight airspeed (at 15,000 feet) and collecting data at 50 knot increments. Tests were conducted at seven specific trim points as shown in table 2. At each test point, the aircraft was stabilized (trimmed) and an aileron-rudder sequential doublet and elevator doublet inputs were made by the pilot. (These tests did not include phugoid maneuvers.)

The total flight test time required for these tests was 13 minutes. Using conventional flight test procedures, the time required to obtain the flight data for determining the same specification requirements is estimated to be 112 minutes. Thus, if phugoid test data are not required, the reductions in flight time are even more dramatic. This saving in flight time is due to the reduction in the time required to conduct the maneuvers and the time required to establish a fewer number of precise trim points.

A similar set of tests was conducted during the Technical Evaluation of the F-14A airplane.[5] These tests are summarized in table 3. As previously demonstrated, a significant reduction in flight test time was achieved using the system identification approach to obtaining flight data (a reduction of 32 to 8 maneuvers was achieved during these tests). Total test time was 8 minutes.

To demonstrate the accuracy of this approach to flight testing, these F-14A data were used to estimate the longitudinal short period characteristics. The body fixed axis system equations used in this analysis are presented below. Elements of the state equation are:

State Vector $\quad x = [\alpha, u, q, \theta]^T \qquad (33)$

Control Vector $\quad \delta = [\delta_e, 1]^T \qquad (34)$

$$F = \begin{bmatrix} Z_\alpha & Z_u & (Z_q + 1) & g \sin\theta_o \\ X_\alpha & X_u & (X_q - \alpha_o u_o) & -g\cos\theta_o \\ M_\alpha & M_u & M_q & 0 \\ 0 & 0 & 1 & 0 \end{bmatrix} \qquad (35)$$

$$G = \begin{bmatrix} Z_{\delta_e} & Z_o \\ X_{\delta_e} & X_o \\ M_{\delta_e} & M_o \\ 0 & Q_o \end{bmatrix} \qquad (36)$$

where Z_o, X_o, M_o, and Q_o are initial conditions

$$\Gamma = [0] \qquad (37)$$

The measurement equations are:

Measurement Vector $\quad y = [\alpha_m, u_m, q_m, \theta_m, a_{z_m}]^T \qquad (38)$

Bias Vector $\quad b = [b_\alpha, b_u, b_q, b_\theta, b_{a_z}]^T \qquad (39)$

Random Measurement Error Vector $\quad \nu = [\nu_\alpha, \nu_u, \nu_q, \nu_\theta, \nu_{a_z}]^T \qquad (40)$

$$H = \begin{bmatrix} K_\alpha & 0 & -K_\alpha l_\alpha / u_o & 0 \\ 0 & 1 & 0 & 0 \\ 0 & 0 & 1 & 0 \\ 0 & 0 & 0 & 1 \\ H(5,1) & H(5,2) & H(5,3) & 0 \end{bmatrix} \qquad (41)$$

where

$$H(5,1) = -u_o Z_\alpha - l_x M_\alpha \qquad (42)$$

$$H(5,2) = -u_o Z_u - l_x M_u \qquad (43)$$

$$H(5,3) = -u_o Z_q - l_x M_q \qquad (44)$$

$$D = \begin{bmatrix} 0 & 0 \\ 0 & 0 \\ 0 & 0 \\ 0 & 0 \\ D(5,1) & D(5,2) \end{bmatrix} \qquad (45)$$

where

$$D(5,1) = -u_o Z_{\delta_e} - l_x M_{\delta_e} \qquad (46)$$

$$D(5,2) = -u_o Z_o - l_x M_o \qquad (47)$$

A general form of the accelerometer measurement equation (fifth row of the H and D matrices) is given by:

$$a_{z_m} = a_{z_T} - l_x \dot{q} + b_{a_z} + \nu_{a_z} \qquad (48)$$

Thus, using these equations in the identification algorithm, this set of data was analyzed assuming that the short period and phugoid modes are uncoupled ($X_u = X_\alpha = X_q = X_{\delta_e} = Z_u = M_u = 0$). The results from these tests are presented in figures 4 and 5.

Figure 4 presents the F-14A stability and control derivative estimates and associated estimate confidence bounds. Figure 5 presents the short period characteristics computed from these derivative estimates and compares them with conventional flight test results. As shown, excellent agreement is obtained between the conventional results and the parameter identification results (conventional results are based on classical hand measurement techniques). There is a significant point to make here in that there is no comparison of conventional flight test determined n_z/α with the parameter identification result because it would have required another test to obtain the conventional data. A sample time history match is presented in figure 6.

The examples presented demonstrate the accuracy and feasibility of using system identification techniques to improve the efficiency of stability and control flight testing. Of course, the examples presented here represent only a portion of the stability and control testing requirements and thus savings in flight time will not be as dramatic when considering the total testing requirements. A previous survey of Navy stability and control flight test programs conducted at NATC indicated that 70 - 90 percent of the testing in large scale programs is devoted to specification testing.[6] Using system identification, it is estimated that 20-30 percent of this portion of the flight program could be eliminated.

Verification of S-3A Power Approach Characteristics

In order to improve the carrier suitability of the S-3A airplane, a program was initiated by the Naval Air Systems Command to identify the origins of any S-3A carrier approach difficulties and to solve them. A portion of this program was to apply the advanced system identification techniques being developed at NATC to S-3A power approach flight test data and compare the resulting characteristics with aerodynamic data the airframe contractor is using to describe the airplane. This verification of the S-3A data base was accomplished by comparing original contractor data with NATC and airframe contractor parameter identification results. (The airframe contractor was conducting a parameter identification analysis concurrent with the independent NATC effort.[7]) Results from the NATC analysis have been published and are summarized below.[8]

The state space model used in this analysis takes on the following form:

Equation of Motion:

State Vector $\quad x = [p, r, \beta, \phi]^T \qquad (49)$

Control Vector $\quad \delta = [\delta_a, \delta_R, 1, \delta_{sp}]^T \qquad (50)$

$$F = \begin{bmatrix} L'_p & L'_r & L'_\beta & 0 \\ N'_p & N'_r & N'_\beta & 0 \\ Y_p + \tan\alpha_o & Y_r - 1 & Y_\beta & \dfrac{g\cos\theta_o}{u_o} \\ 1 & \tan\theta_o & 0 & 0 \end{bmatrix} \qquad (51)$$

$$G = \begin{bmatrix} L'_{\delta_a} & L'_{\delta_R} & L'_o & L'_{\delta_{sp}} \\ N'_{\delta_a} & N'_{\delta_R} & N'_o & N'_{\delta_{sp}} \\ Y_{\delta_a} & Y_{\delta_R} & Y_o & Y_{\delta_{sp}} \\ 0 & 0 & \phi_o & 0 \end{bmatrix} \qquad (52)$$

where L_o, N_o, Y_o, and ϕ_o are initial conditions and primed derivatives are defined as:

$$N'_x = \frac{N_x + \left(\frac{I_{xz}}{I_{zz}}\right) L_x}{1 - \frac{I_{xz}^2}{I_{xx} I_{zz}}} \quad (53)$$

and:

$$L'_x = \frac{L_x + \left(\frac{I_{xz}}{I_{xx}}\right) N_x}{1 - \frac{I_{xz}^2}{I_{xx} I_{zz}}} \quad (54)$$

where the subscript (x) denotes p, r, β, δ_a, δ_R, and δ_{sp}. In addition, an aerodynamic spoiler lag was applied to spoiler measurements.

$$\dot{\delta}_{sp_{LAG}} = -\left(\frac{1}{\tau}\right)\delta_{sp_{LAG}} + \left(\frac{1}{\tau}\right)\delta_{sp_M} \quad (55)$$

where:

$$\tau = \frac{15}{V_{True\,(in\ knots)}} \quad (56)$$

This correction is required, especially at low speeds, because of the time required for lift to build up after a spoiler input (aerodynamic lag). The solution of equation (55) ($\delta_{sp_{LAG}}$) was used as the input for spoiler (equation (50)).

Measurement Equations:

Measurement Vector $y = \left[P_m, r_m, \beta_m, \phi_m, a_{y_m}\right]^T$ (57)

Bias Vector $b = \left[b_p, b_r, b_\beta, b_\phi, b_{a_y}\right]^T$ (58)

Random Measurement Error Vector $\upsilon = \left[\upsilon_p, \upsilon_r, \upsilon_\beta, \upsilon_\phi, \upsilon_{a_y}\right]^T$ (59)

$$H = \begin{bmatrix} 1 & 0 & 0 & 0 \\ 0 & 1 & 0 & 0 \\ \frac{-k_\beta I_z}{u_o} & \frac{k_\beta I_x}{u_o} & k_\beta & 0 \\ 0 & 0 & 0 & 1 \\ H(5,1) & H(5,2) & H(5,3) & 0 \end{bmatrix} \quad (60)$$

where:

$$H(5,1) = u_o Y_p - I_Z L'_p + I_x N'_p \quad (61)$$

$$H(5,2) = u_o Y_r - I_Z L'_r + I_x N'_r \quad (62)$$

$$H(5,3) = u_o Y_\beta - I_Z L'_\beta + I_x N'_\beta \quad (63)$$

$$D = \begin{bmatrix} 0 & 0 & 0 & 0 \\ 0 & 0 & 0 & 0 \\ 0 & 0 & 0 & 0 \\ 0 & 0 & 0 & 0 \\ D(5,1) & D(5,2) & D(5,3) & D(5,4) \end{bmatrix} \quad (64)$$

where:

$$D(5,1) = u_o Y_{\delta_a} - I_Z L'_{\delta_a} + I_x N'_{\delta_a} \quad (65)$$

$$D(5,2) = u_o Y_{\delta_R} - I_Z L'_{\delta_R} + I_x N'_{\delta_R} \quad (66)$$

$$D(5,3) = u_o Y_o - I_z L'_o + I_x N'_o \quad (67)$$

$$D(5,4) = u_o Y_{\delta_{sp}} - I_z L'_{\delta_{sp}} + I_x N'_{\delta_{sp}} \quad (68)$$

A general form of the lateral acceleration measurement equation can be written as:

$$a_{y_m} = a_{y_T} - \dot{p} I_z + \dot{r} I_x + b_{a_y} + \upsilon_{a_y} \quad (69)$$

Parameter identification analysis results are presented in table 4 and figures 7 and 8. As shown, final airframe contractor and NATC identification results are in general agreement but show some significant differences with the original data. Based on this comparison and the time history matches presented in figure 8, it was concluded that the original data base did not accurately represent the lateral-directional aerodynamics of the S-3A. Thus, the NATC parameter identification analysis essentially verifies the final airframe contractor aerodynamic data base of the S-3A (excluding spoiler sideforce characteristics).

As with the use of any new technology, additional comparisons are desirable for obtaining confidence in experimental results. These comparisons are presented in table 5 and figures 9 and 10. Table 5 presents a comparison of lateral-directional modal characteristics (stability and control specification requirements). Figure 9 presents a comparison of parameter identification results and parameter estimates based on a flight test differential engine thrust sideslip maneuver. Figure 10 is a comparison of flight test results for a steady heading sideslip maneuver (lateral-directional static stability characteristics) and parameter identification results. These three comparisons show good agreement between the parameter identification and conventional flight test results and in this light verify the parameter identification estimates.

EA-6B Catapult Launch Capabilities

A program was initiated by the Naval Air Systems Command to determine EA-6B catapult launch capabilities as limited by an engine failure immediately following launch. Part of this program was to determine from flight tests the longitudinal aerodynamic characteristics of this airplane for use in a catapult launch simulation. This simulation will be used to define critical flight areas prior to any actual flight tests. The results from this program are to be published and are summarized below.[9]

The state equations used in the analysis of the F-14 airplane were augmented in this case to include a lag on the angle of attack vane.

$$\dot{\alpha}_S = -\left(\frac{1}{\tau_\alpha}\right)\alpha_S + \left(\frac{1}{\tau_\alpha}\right)\alpha_T - \left(\frac{1}{\tau_\alpha}\right)\left(\frac{l_\alpha}{u_o}\right)q \tag{70}$$

The measurement equation for the angle of attack vane was then modified as follows:

$$\alpha_m = K_\alpha \alpha_S + b_\alpha + n_\alpha \tag{71}$$

Parameter identification estimates for the stability and control derivatives and comparisons with conventional flight test results and wind tunnel data are presented in figure 11 and table 6. As shown in figure 11, reasonable agreement between the Z force derivatives was obtained; however, there is a large difference in the estimates for the pitching moment derivatives. The largest difference is in the pitch damping derivative ($C_{m_q}+C_{m_{\dot{\alpha}}}$) which would indicate a significantly higher level of short period damping than is obtained from the wind tunnel data. (This is specifically pointed out in the following paragraphs.) For the phugoid mode, close agreement is obtained between the parameter identification estimate and conventional flight test results for the parameters C_{x_u} and C_{z_u} . Representative time history comparisons for short period and phugoid aircraft responses are presented in figures 12 and 13.

Data are now presented to demonstrate that these stability and control derivative estimates can be used to determine compliance with dynamic and static longitudinal stability specification requirements. Comparisons of dynamic stability characteristics are presented in table 7 and show good agreement between parameter identification and conventional flight test results for both the phugoid and short period modes. However, comparison with wind tunnel data shows that short period damping estimates are approximately one-half of the parameter identification and conventional flight test results.[10] As previously discussed, this would indicate that the wind tunnel value for the pitch damping derivative is low. Comparisons of static longitudinal characteristics in figure 14 show good agreement between conventional results and the elevator to airspeed gradient computed using the parameter identification stability derivative estimates.[11]

Integrated Flight Testing

This paper has dealt with the savings that can be realized in stability and control flight testing by utilizing system identification technology; however, an even more dramatic reduction in flight test costs and time could be achieved by an integrated approach to flight testing. This integrated flight testing will be made possible by a further growth in data analysis technologies, such as system identification and dynamic performance. This can be readily illustrated by considering Navy vehicle dynamics tests which consist of:

a. Aerodynamics.

b. Stability and control.

c. Performance.

d. Automatic flight control system.

e. Automatic carrier landing system.

f. Propulsion.

g. Structures and flutter.

Under current test philosophy, these tests require at least four airplanes devoted to the flight tests development program. Considering the state of current data analysis technology in system identification and dynamic performance, it is possible to integrate these test requirements to form a reduced flight program (i.e., a reduction in both flight tests and required aircraft could be achieved). This is illustrated by considering the flight profile in figure 15 which consists of primarily a series of level flight accelerations/decelerations and constant Mach climbs and descents.[12] The aircraft is stabilized at various points during the acceleration/deceleration maneuvers and various maneuvers are performed to collect data for military specification compliance determination. For example, if these maneuvers consisted of an elevator doublet, aileron-rudder doublet, phugoid, and engine Bodie, then the majority of stability and control, performance, propulsion, and automatic stabilization requirements could be determined using advanced data analysis techniques. (The acceleration/deceleration data are used in dynamic performance methods to estimate aircraft performance characteristics.) Although the examples used do not cover all of the test requirements for these areas, such as specialized automatic control functions, the tests conducted in this flight profile could be easily modified to include them. (This would extend the number of flights required to complete the flight profile for one aircraft loading and configuration.)

Concluding Remarks

The use of system identification technology to provide for more effective aircraft stability and control flight testing has been demonstrated. This is accomplished by either improving the efficiency of the flight test and/or providing for a more comprehensive data analysis. Thus, the application of this technology to flight testing provides for an in-depth understanding of the cause and effect relationship in aircraft stability and control characteristics. The one remaining objective to be reached in this program is to apply this technology in a large scale flight program to update the aerodynamic data base of an aircraft throughout its flight envelope. To meet this objective, current plans call for the application of this technology in the development programs for automatic carrier landing systems, TA-4KU aircraft, HARRIER, and the Navy's new fighter, the F-18.

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Table 1 Reductions in Stability and Control Flight Testing Using Parameter Identification

Test	Number of Maneuvers	
	Conventional	*Parameter* Identification
Longitudinal		
o Short Period Damping (ζ)	1	2
o n_z/α	1	
o Phugoid	1	
o Static Stability	1 (7 Test Points)	
Lateral-Directional		
o Dutch Roll (ζ, ω)	1	1
o Spiral Mode ($1/\tau_s$)	1	
o Roll Mode ($1/\tau_R$)	1	
o Dutch Roll Coupling and Excitation	1	
o Static Stability	1 (7 Test Points)	
Total Maneuvers	9	3
Estimated Flight Time	20 min	5 min

Table 2 Parameter Identification
TA-4J Stability and Control Flight Tests

Trim Airspeed (KEAS)	Number of Maneuvers (1)	Flight Time
165	2	13 minutes
200	2	
250	2	
300	2	
350	2	
400	2	
V_{MAX}	2	
	Total 14	

NOTE: (1) Conventional tests would have required 56 maneuvers and 112 minutes.

Table 3 Parameter Identification
F-14A Stability and Control Flight Tests

Trim Airspeed (KEAS)	Number of Maneuvers (1)	Flight Time
250	2	8 minutes
350	2	
450	2	
500	2	
	Total 8	

NOTE: (1) Conventional testing would have required 32 maneuvers.

Table 4 Comparison of S-3A Power Approach Parameter Identification Results with Orignial Contractor/Wind Tunnel Data

No.	Parameter	Wind Tunnel Data	Airframe Contractor Identification Results	NATC Identification Results			
				Parameter Estimate	Parameter Estimate Confidence Bounds (2σ)(3)	Parameter Estimate +2σ	Parameter Estimate -2σ
1	L'_p	-1.683	-1.956	-1.855	0.1326	-1.988	-1.722
2	L'_r	0.381	1.137	1.113	0.0870	1.200	1.026
3	L'_β	-2.065	-2.957	-2.825	0.2046	-3.0296	-2.6204
4	N'_p	-0.606	-0.490	-0.526	0.0578	-0.5834	-0.4682
5	N'_r	-0.302	-0.103	-0.1299	0.0318	-0.1617	-0.0981
6	N'_β	0.837	0.966	0.9319	0.0848	1.0167	0.8471
7	Y_p	-0.0026	-0.0026	-0.0026	-	-	-
8	Y_r	0.0075	0.0075	0.0384	0.002	0.0404	0.0364
9	Y_β	-0.121	-0.113	-0.1306	0.0022	-0.1328	-0.1284
10	L'_{δ_R}	0.449	0.449	0.287	0.0752	0.3622	0.2118
11	N'_{δ_R}	-0.799	-0.799	-0.795	0.0268	-0.8218	-0.7687
12	Y_{δ_R}	0.030	0.030	0.0345	0.0064	0.0409	0.0281
13	L'_{δ_a}	0.596	0.596	0.596	-	-	-
14	N'_{δ_a}	0.075	0.075	0.075	-	-	-
15	Y_{δ_a}	0	0	0	-	-	-

NOTES: (1) Body fixed axis system
(2) Data linearized about trim angle of attack (approximately 8 degrees)
(3) 2σ confidence bounds represent a 95% confidence interval
(4) Clean loading

Table 5 S-3A Power Approach Modal Characteristics

Modal Characteristics	Conventional Results (1)	Wind Tunnel Data	Airframe Contractor Identification Results	NATC Identification Results
Spiral Mode Time Constant (sec) (2)	Slightly Divergent to Neutral	157.73	17.00	18.1
Roll Mode Time Constant (sec) (3)	-0.51	-0.546	-0.521	-0.539
Dutch Roll Frequency (rad/sec)	1.417	1.287	1.405	1.412
Dutch Roll Damping	0.10	0.0998	0.102	0.113

NOTES: (1) Hand measurement techniques
(2) Positive sign indicates divergent spiral mode
(3) Negative sign indicates convergent roll mode
(4) Clean loading

Table 6 EA-6B Phugoid Parameter Identification Results in the Power Approach Configuration(1)

Parameter	Parameter Estimate (4)	Parameter Estimate Confidence Bound (2σ)	Conventional Flight Test Estimate (3)
C_{x_u}	-0.231	0.006	-0.281
C_{z_u}	-1.523	0.022	-1.632
C_{m_u}	-0.029	0.002	-
C_{x_α}	+0.271 (2)	-	-
C_{x_q}	0 (2)	-	-
$C_{x_{\delta_e}}$	0 (2)	-	-

NOTES: (1) Loading - 3 ECM Pods/2 Tanks
(2) Wind tunnel data (not estimated)
(3) Estimates are based on flight test drag polar data
(4) Body fixed axis system

Table 7 EA-6B Frequency and Damping Characteristics in the Power Approach Configuration(1)

Modal Parameter		Parameter Identification Result	Conventional Flight Test Result	Wind Tunnel Estimate
Short Period	$\omega_{n_{sp}}$ (rad/sec)	1.071	1.406	1.178
	ζ_{sp}	0.630	0.670	0.345
Phugoid	ω_{n_p} (rad/sec)	0.148	0.147 (2)	0.147 (2)
	ζ_p	0.092	0.093 (2)	0.093 (2)

NOTES: (1) Loading - 3 ECM Pods/2 Tanks
(2) Estimate based on combination of wind tunnel and conventional flight test results

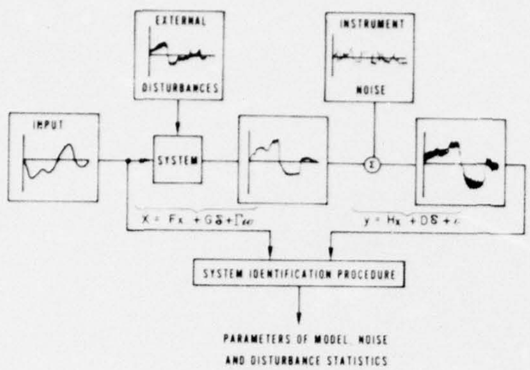

Figure 1 Parameter Identification Procedure

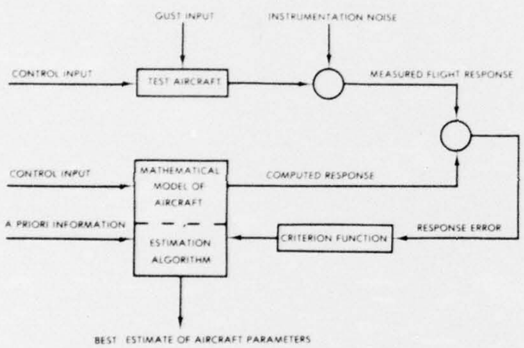

Figure 2 Iterative Parameter Identification Process

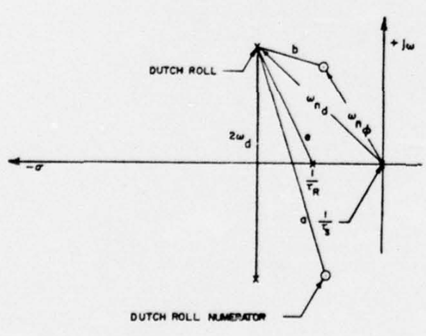

Figure 3 Root Locus Plot of the Roll Rate to Aileron Transfer Function

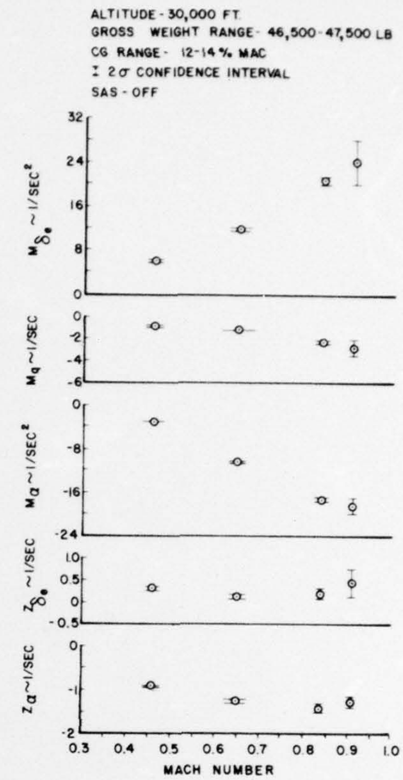

Figure 4 F-14A Short Period Stability and Control Derivative Estimates in Configuration Cruise

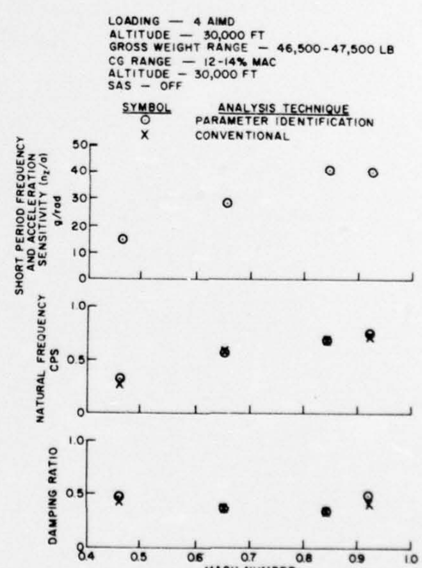

Figure 5 F-14A Short Period Stability and Control Characteristics in Configuration Cruise

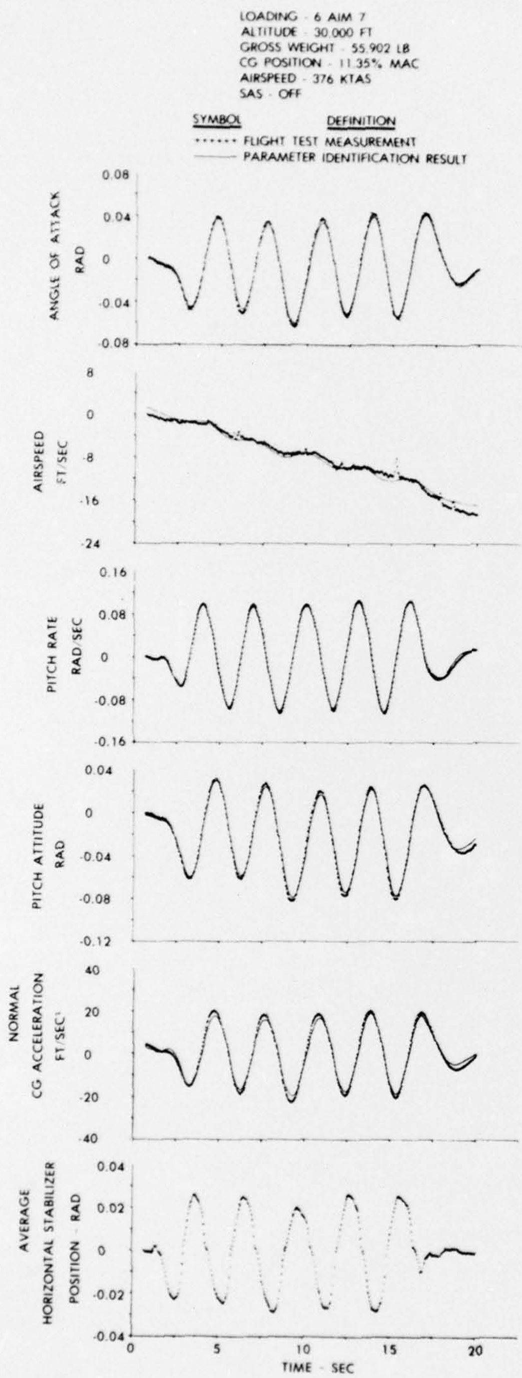

Figure 6 F-14A Short Period Time History Comparison in Configuration Cruise

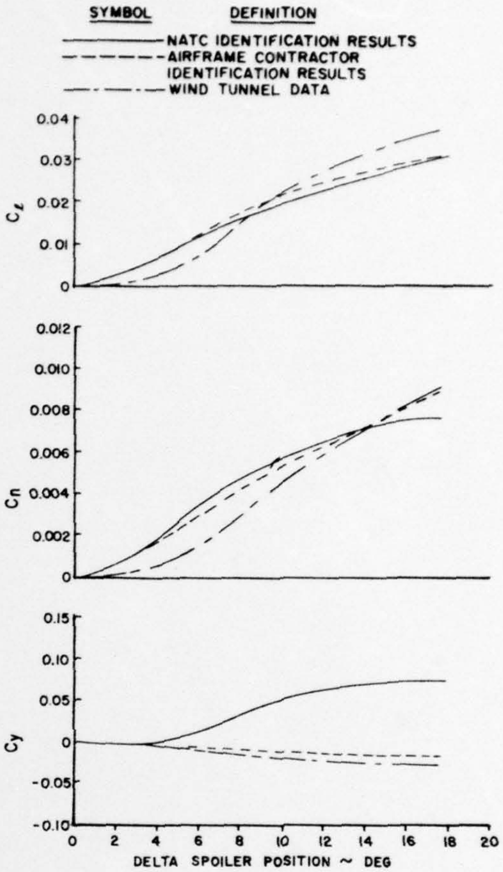

Figure 7 S-3A Power Approach Spoiler Coefficients as a Function of Spoiler Position for Body Axis

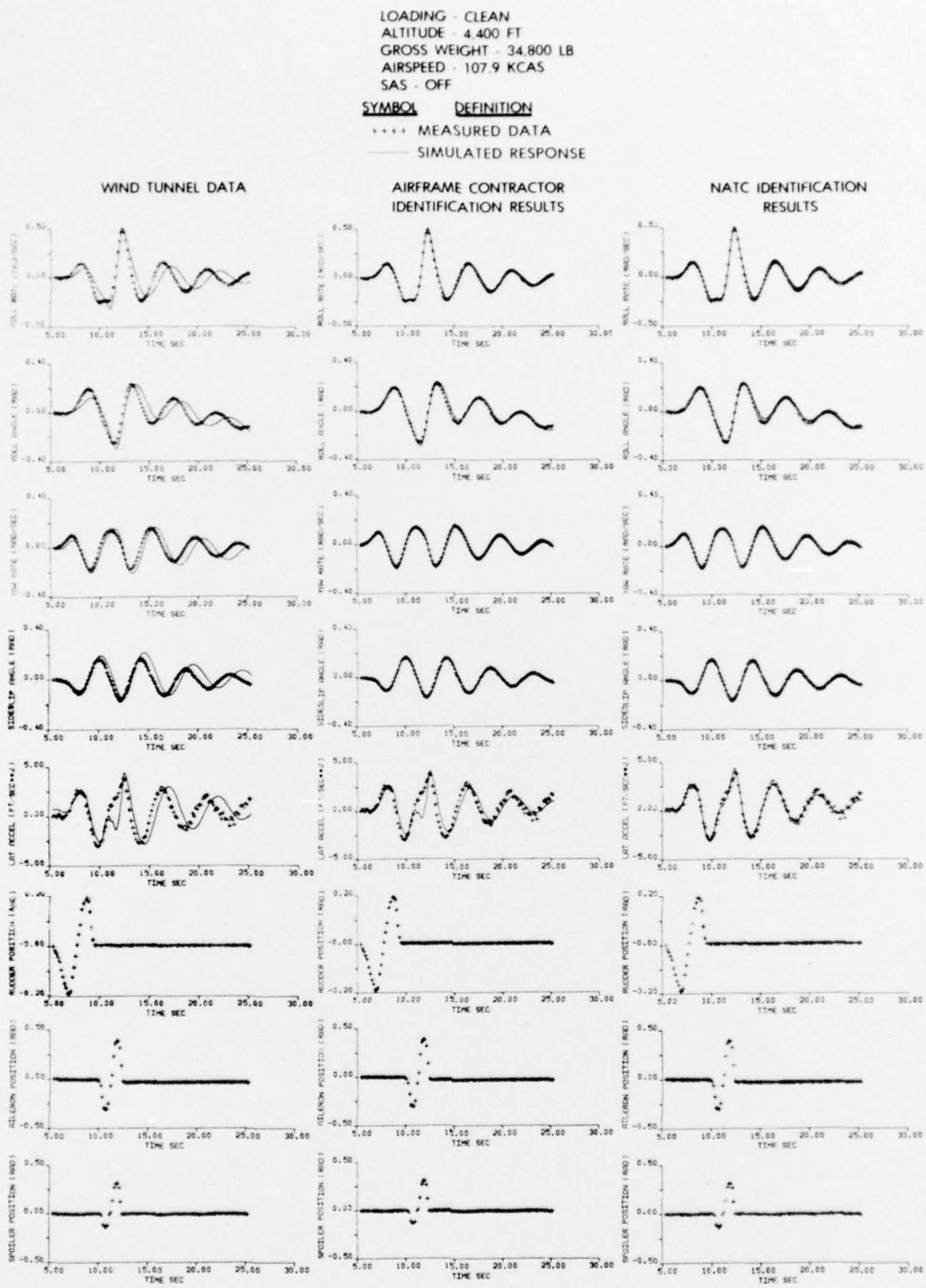

Figure 8 Time History Comparison of S-3A Power Approach Parameter Identification Results with Original Contractor/Wind Tunnel Data

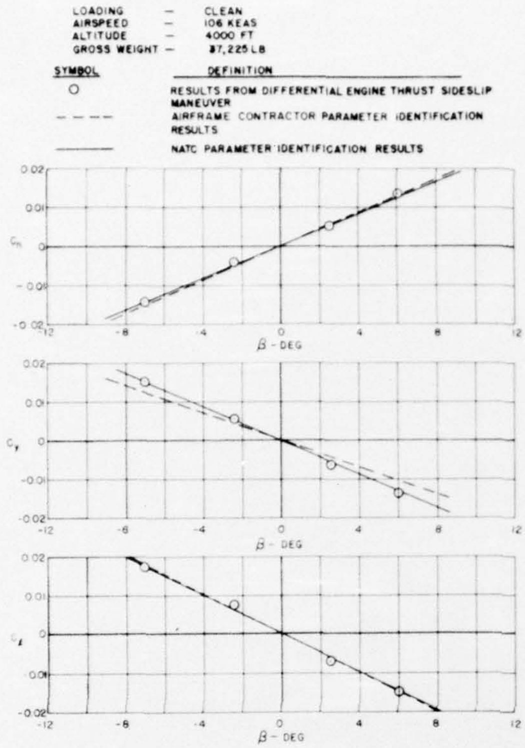

Figure 9 Comparison of S-3A Power Approach Parameter Identification Results and Parameter Estimates from Differential Engine Thrust Sideslip Maneuver

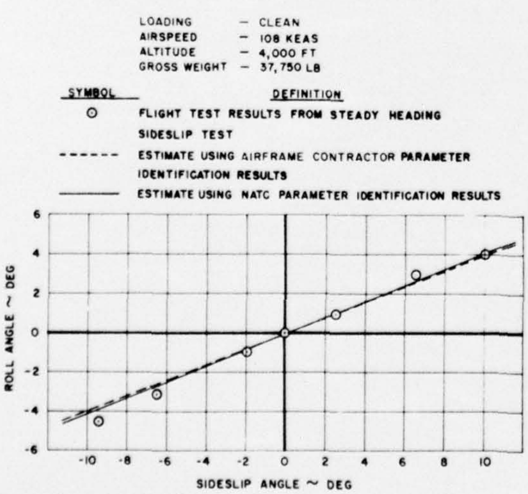

Figure 10 Comparison of S-3A Power Approach Flight Test Results for Steady Heading Sideslip and Parameter Identification Results

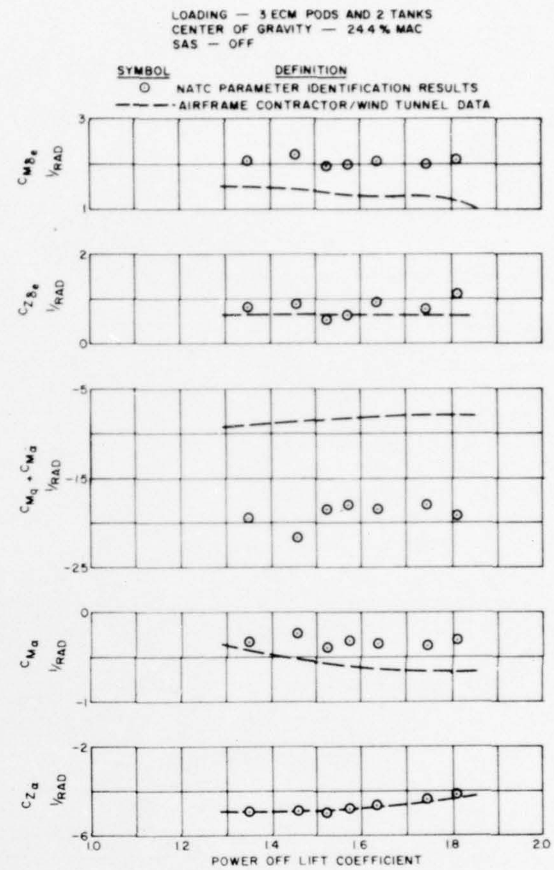

Figure 11 EA-6B Power Approach Short Period Parameter Identification Results

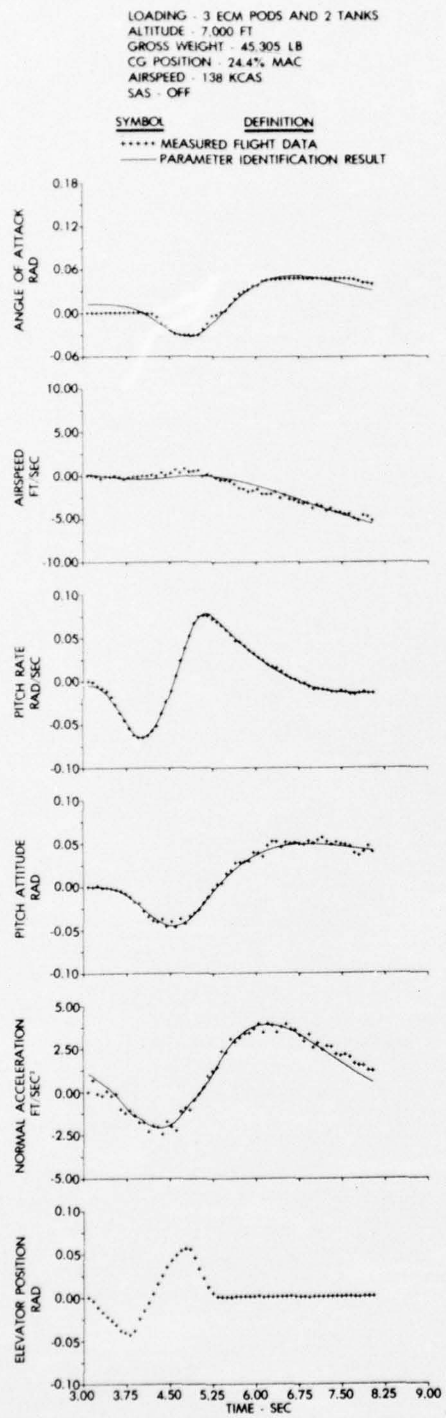

Figure 12 EA-6B Power Approach Short Period Time History Comparison

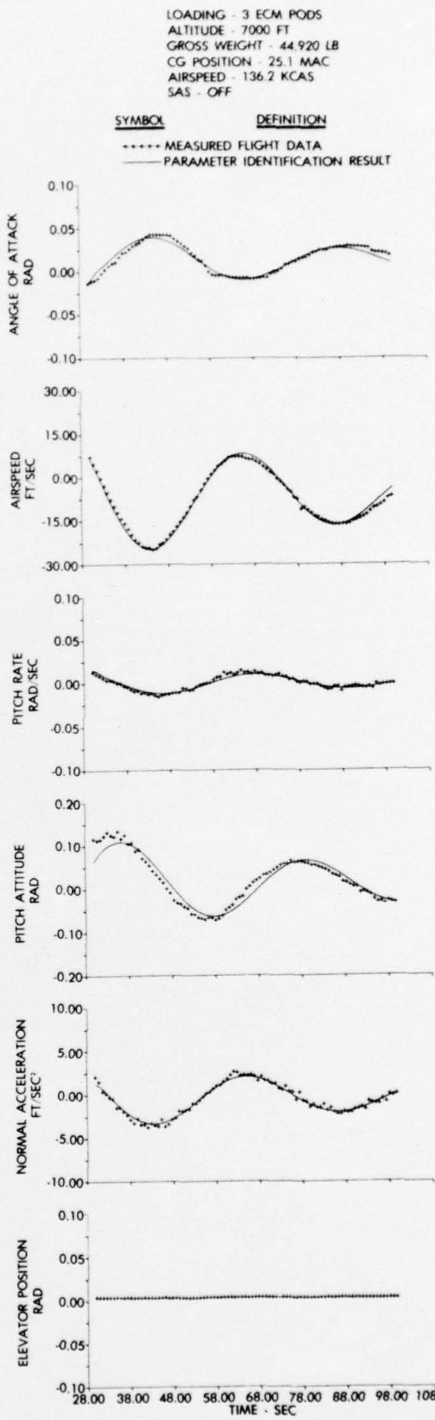

Figure 13 EA-6B Power Approach Phugoid Time History Comparison

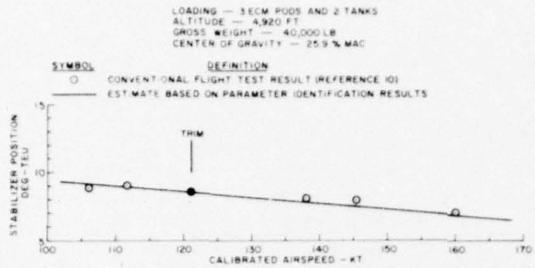

Figure 14 Comparison of EA-6B Power Approach Static Longitudinal Stability Characteristics

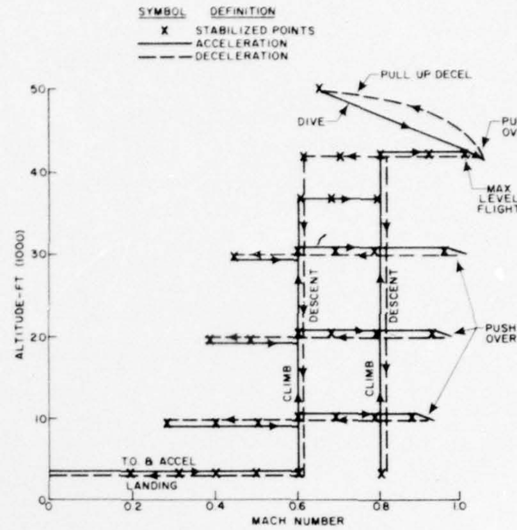

Figure 15 Example of Flight Profile for Integrated Vehicle Dynamics Flight Testing

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20. ABSTRACT (Continue on reverse side if necessary and identify by block number)

The development of system identification technology was undertaken to provide for more effective aircraft flight testing by reducing the time required to conduct specific tests and/or to provide for a more comprehensive data anlaysis. F-14A and TA-4J flight test results presented demonstrate that the flight time required to obtain stability and control data can be significantly reduced without loss in accuracy of conventional flight test derived parameters. Presentation of S-3A and EA-6B system identification results demonstrate that this technology can be successfully used to update the aerodynamic → next page

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data bases of modern jet aircraft from flight test data. These system identification results are compared with wind tunnel data and flight test derived parameters to demonstrate the accuracy of this new technology. Applications of this technology to integrate several areas of aircraft flight testing are discussed.